Aeroelastic studies
on a high-performance
swept-wing airplane

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"AEROELASTIC STUDIES ON A HIGH-PERFORMANCE SWEPT-WING AIRPLANE"

by: C. Desmond Pengelley* and David Benun+

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*Southwest Research Institute
San Antonio, Texas

+Guggenheim Aeronautical Laboratory
California Institute of Technology
Pasadena, California

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ABSTRACT OF PAPER TITLED

AEROELASTIC STUDIES ON A HIGH-PERFORMANCE SWEPT-WING AIRPLANE

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Comprehensive aeroelastic analyses and wind tunnel tests have been conducted on a model of a typical high-performance airplane with swept surfaces. The investigation covered the effect of dynamic pressure upon steady state longitudinal control and stability characteristics as well as aileron effectiveness.

The analysis was based upon a superposition method which yields a convergent series which may be applied to any structure on which the load is a function of the deflection. It has been shown that after a few terms this series degenerates to an elementary geometric progression which may be summed to infinity. This conclusion is applicable even to cases where the aeroelastic effects are large and the structure is actually approaching a condition of instability.

Wind tunnel tests were carried out both at the Massachusetts Institute of Technology and the California Institute of Technology. Measured data were compared with the results of the theoretical analysis.

INTRODUCTION

The methods presented herewith were originally* developed in an attempt to check certain aerodynamic phenomena which were encountered on an airplane during high-speed flight and suspected of being caused by the elasticity of the structure. The results of an analysis, based on this method, essentially confirmed these suspicions.

Since that time extensive wind tunnel tests have been made on a complete aeroelastic model of an airplane with highly swept wing and tail surfaces. The model also incorporated a flexible fuselage.

The nature of the method is such that the vast majority of analysis on an airplane having flexible parent surfaces and also flexible control surfaces can be carried out before it is necessary to specify the magnitudes of important variable parameters such as dynamic pressure (q), angle of attack (α), control surface deflection (δ), induced angle of attack at tip due to damping in roll or pitch (D), load factor (N), and any other aerodynamic parameters resulting from airplane attitude or control configuration.

The analysis will yield certain constants (A_{MB1}, A_{MX2}; D_B etc.) which are fixed for a given structure (i.e., wing or tail) and are independent of the magnitudes of the foregoing parameters. These constants are

*Other treatments of this problem are given in References (1) thru (5). The method of Reference (3), although confining the problem to an unswept wing and attacking it from a slightly different point of view, yields expressions for lift coefficient and distribution which are the same as those obtained from this work.

then inserted into appropriate equations to obtain the total lift, moment, etc., as simple functions of the aforementioned variable parameters. Examples are shown following Eq. (14). The number of terms will vary with the structure but in most cases will probably be between three and six. The A and D constants which depend only upon planform and structure may be placed in Eq. (14) to determine the moment coefficient for any chosen flight condition. Other aerodynamic coefficients may be established in a similar manner. No assumptions are required regarding deflection shapes since these are one of the end products of the analysis.

From the above mentioned data, trim, stability, control, reversal, and divergence characteristics may be obtained for any condition of steady accelerated flight. In addition, a method has been outlined which is particularly convenient for treating fuselage flexibility.

The analytical methods set forth herewith are based upon Reference (6), which covers in detail the computational procedures which may be applied to actual airplanes. The comparison of analytical results with test data is taken from References (7) and (8).

As a result of this investigation, it is specifically recommended that full span ailerons, split into inboard and outboard sections, should be tried for use on high aspect ratio flexible airplanes such as the one shown in Figure 3. Both sections would be used as a single aileron at low speeds where aeroelastic effects are small and large values of (pb/2v) are essential. As the reversal speed of the outboard aileron is approached, however, a suitable mechanism would be used to make it inoperative. Since the inboard section would have a much higher reversal speed, it would, therefore, provide positive control at higher speeds than would be otherwise possible even though at low speeds it would not provide a sufficiently high value of (pb/2v).

METHOD

Consider a lifting surface of arbitrary planform (Figure 1), its displacement from any reference plane being represented by the coordinate, z. It is assumed that for any known fixed distribution of z the pressure distribution can be determined, and for any known fixed pressure distribution the resulting elastic distortions can be obtained. In addition, linearity is assumed for structural deflections and aerodynamic induction, as a result of which superposition of loading is valid.

Before elastic distortion is allowed to occur, let z_o represent the deflection pattern. This initial distribution may be defined as the "Basic Configuration" and produces a lift (L_o) and moment (M_o) on the entire surface proportional to q, the dynamic pressure and z_{ot}, the value of z_o at some reference point. However, the pressure distribution will be independent of both q and z_{ot}. Thus,

$$L_o = q\, z_{ot}\, k_{Lo} \qquad (1)$$

$$M_o = q\, z_{ot}\, k_{Mo} \qquad (2)$$

where k_{Lo} and k_{Mo} are independent of q and z_{ot} and are functions only of the basic configuration which constitutes a known initial aerodynamic shape, and may, therefore, be computed.

If the surface is allowed to distort under the above loading, the first "incremental deflection" (z_1) will occur, such that:

$$z_{1t} = K_1 M_o \tag{3}$$

The distribution of z_1 depends upon the structure only whereas its magnitude is also proportional to M_o (or L_o). Thus, K_1 may be evaluated. This deflection will produce the first "elastic induced" loading (L_1), so that

$$L_1 = q\, z_{1t}\, k_{L1} \tag{4}$$

Similarly $$M_1 = q\, z_{1t}\, k_{M1} \tag{5}$$

The above loads result in a second incremental deflection z_2, which, in turn, produces a second induced elastic loading. This procedure of obtaining successive elastic-induced loadings can be carried out indefinitely and in general

$$z_{nt} = K_n M_{(n-1)} \tag{6}$$

$$L_n = q\, z_{nt}\, k_{Ln} \tag{7}$$

$$M_n = q\, z_{nt}\, k_{Mn} \tag{8}$$

where the constants k_{Ln}, k_{Mn}, and K_n may be evaluated for all values of "n" for any known structure, planform, and basic configuration.

The following notation shall be used:

$$k_{\underline{Mn|}} = k_{M1} \times k_{M2} \times k_{M3} \times \ldots \times k_{Mn}$$

$$K_{\underline{n|}} = K_1 \times K_2 \times K_3 \times \ldots \times K_n$$

It can then be shown that

$$M_n = q^n \, k_{\underline{Mn}|} \, K_{\underline{n}|} \, M_o \tag{9}$$

The total equilibrium moment acting on the flexible surface due to any basic configuration F may be obtained by summing the initial and all subsequent elastic-induced loadings as follows:

$$M_F = \sum_{0}^{\infty} M_n \tag{10}$$

In some cases where convergence is rapid, numerical values of M_F may be obtained by summing Eq. (10) for only a limited number of values of n; however for certain aeroelastic surfaces, for example, highly swept wings, the nature of the elastic-induced loadings is such that successive terms in the expression may alternate from positive to negative with increasing magnitude. Needless to say, such a series is nonconvergent in the conventional sense though it may actually have a finite sum. It is, therefore, necessary to resort to the utilization of certain fortunate characteristics of flexible structures, which will be set forth herewith.

CONCEPT OF "CHARACTERISTIC SHAPE"

A "Characteristic Shape" is defined as follows:

> When a certain pattern of structural deflections produces an "Elastic-Induced Loading" which is of such a nature that it, in turn, produces an incremental deflection pattern which has the same shape as the initial pattern, then this particular distribution of deflection is defined as a "Characteristic Shape" and the corresponding loading distribution is defined as a "Characteristic Loading."

The following hypotheses* regarding "Characteristic Shapes" are stated without proof:

(1) There is at least one+ "Characteristic Shape" for any given structure in a linear system.

(2) Successive "Elastic-Induced Loadings" will converge to a "Characteristic Loading."

APPLICATION OF CHARACTERISTIC SHAPES

From the foregoing hypotheses, let it be assumed that a "Characteristic Shape" has been reached at the mth elastic-induced loading so that all subsequent loadings will have the same shape within the desired degree of accuracy.

It may be shown that the sum of the elastic-induced loadings, from the "Characteristic Loading" on, inclusive, becomes

$$\sum_{n=m}^{\infty} M_{(m+n-1)} = \left[\frac{q^{m}\, k_{Mm}\, K_{m}\, M_{o}}{1-q\, k_{Mm}\, K_{(m+1)}}\right] \tag{11}$$

*The first hypothesis can be rigorously proved mathematically within the assumption of the linearized theory. The writers have not produced a formal general proof for the second hypothesis but believe intuitively that it is true for real structures. In any event, it has been true for all cases and examples that have been investigated by the writers to date.

+In practice there may be an infinite number.

The expression for the total moment on a flexible wing due to a basic configuration "F"* can now be written:

$$M_F = qFa_{MF}cS \left[1 + q^{A_{MF1}} + q^2 A_{MF2} + --- + \frac{q^m A_{MFm}}{1-q^{D_F}} \right] \qquad (12)$$

where $A_{MFn} = k_{\underline{Mn}\rfloor} K_{\underline{n}\rfloor}$ where k and K functions apply specifically to basic configuration "F" for all values of "n" and can be evaluated as a result of the assumptions already specified.

$D_F = k_{Mm} K_{(m+1)}$ where "m" refers to "Characteristic Shape."

$a_{MF} = dC_M/dF$ as $q \longrightarrow o$

A similar expression may be obtained for the total lift except that the subscript "M" should be replaced by "L" and the chord length "c" omitted.

Associated with each elastic-induced loading there is a corresponding nondimensional function "f_n" which defines the deflection shape of the surface. Thus

$$z_n = f_n \, z_{nt}$$

*A basic configuration refers to some known initial distribution of z_o from the zero lift plane such as built in twist (B), additional angle of attack (α), aileron deflection (δ), etc. The symbol "F" is a generalized quantity representing the corresponding angle of attack at some reference station at very low speeds where aeroelastic effects are negligible.

Then, due to a basic configuration "F", the equation for z_F, at some chosen point on the flexible surface, would be as follows:

$$z_F = z_{ot}\, f_{Fo} \left[1 + q\, A_{zF1} + q^2\, A_{zF2} + \text{---} + \frac{q^m\, A_{zFm}}{1 - q^{D_F}} \right] \tag{13}$$

where $A_{zFn} = \left[\frac{(f_{Fn})}{(f_{Fo})} \; \frac{(k_{Mo})}{(k_{Mn})} \; A_{MFn} \right]$ where k and f functions are evaluated specifically for basic configuration "F".

It should be noted that z_F has been used to indicate the linear displacement at any point on an aeroelastic surface and, thus, the solution is general. In practice it is usually far more desirable to check angular deflections only. Since Eq. (13) provides a complete solution for the equilibrium shape of the wing, the spanwise load distribution and induced drag may readily be obtained by conventional rigid wing methods for any chosen flight condition.

APPLICATION TO COMPLETE AIRPLANE

A - Loading Conditions on Aircraft Surface

In practice, the final total pressure distribution on an airplane may result from various combinations of such variables as B, α, δ, N, and D. In order to permit their independent variation, it is desirable to treat them as separate basic configurations. The principle of superposition may then be used to obtain resultant forces. The various constants, (i.e., k_L, k_M, K, etc.) will be different for each configuration (i.e., B, α, δ, N, D) but, once obtained, will be fixed.

The general expression for the total moment coefficient on a flexible surface due to any or all of the five conditions acting simultaneously is as follows:

$$C_M = BC_{MB} + \alpha C_{M\alpha} + \delta C_{M\delta} + DC_{MD} + NC_{MN} \tag{14}$$

where $$C_{MB} = a_{MB}\left[1 + qA_{MB1} + q^2A_{MB2} + \ldots + \frac{q^mA_{MBm}}{1 - q^DB}\right]$$

$$C_{M\alpha} = a_{M\alpha}\left[1 + qA_{M\alpha 1} + q^2A_{M\alpha 2} + \ldots + \frac{q^mA_{M\alpha m}}{1 - q^D\alpha}\right]$$

the expressions for $C_{M\delta}$, C_{MD}, and C_{MN} being similar.

The "a" quantities represent the respective slopes of the pitching moment coefficient curves with regard to the appropriate parameter (i.e., B, α, δ, D, N) at very low speeds as $q \to o$. Except for a_{MN}, the "a" quantities physically represent the slopes of the corresponding moment coefficient curves for a rigid wing. With regard to a_{MN} this is not true since a rigid wing cannot deflect under gravity; and, therefore, no aerodynamic coefficient could be induced by gravity effects. It will also be noted that "a"$_{MN}$ depends upon the structural stiffness of the wing while all other "a" quantities do not. The reason for this is that the "basic configuration" for condition "N" is the result of structural deflections produced by gravity and inertia loads instead of simple aerodynamic shapes established by the designer who may arbitrarily choose the the "built-in-twist," "angle of attack," or "control surface deflection." However, when he chooses the load factor, he leaves it up to the structure to establish for itself the corresponding deflection pattern — i.e., basic configuration.

The angle of attack due to damping "D" is caused by angular velocities due to roll or symmetrical pull out with a highly swept wing.

An expression for the lift coefficient may be obtained which is similar to Eq. (14) as follows:

$$C_L = BC_{LB} + \alpha C_{L\alpha} + \delta C_{L\delta} + DC_{LD} + NC_{LN} \tag{15}$$

where C_{LB}, $C_{L\alpha}$, etc., may be expressed in terms of the slopes of the lift coefficient curves $a_{L\alpha}$, $a_{L\delta}$, etc., when $q \to o$ as was done for moment coefficients following Eq. (14).

Downwash may be expressed in the same way, so that

$$\varepsilon = \left[B\varepsilon_B + \alpha\varepsilon_\alpha + \delta\varepsilon_\delta + D\varepsilon_D + N\varepsilon_N \right] \tag{16}$$

However, due to the relatively poor state of the art pertaining to the analysis of downwash behind even a rigid wing, it is doubtful whether Eq. (16) can be used to much advantage; and it is recommended that some reasonable assumption be made, for example, that the downwash angle is proportional to the lift coefficient.

B - Fuselage Flexibility

For chosen values of α, q, and N, Eqs. (14), (15), and (16) may be used to determine the forces and moments on the wing and the downwash angle behind it. Provided that the fuselage is rigid, the corresponding tail angle of attack may be established in the conventional manner, namely

$$\alpha_{Tr} = \alpha_W + \alpha_i - \varepsilon \qquad \text{(for rigid fuselage)}$$

However, when fuselage flexibility is appreciable, the above equation must be amplified as follows

$$\alpha_T = \alpha_{Tr} + \alpha_f \qquad \text{(for flexible fuselage)}$$

The quantity α_f, which is the change in tail incidence due to fuselage deflection, depends upon the tail loads which, in turn, depend upon the tail angle of attack. This problem may be treated conveniently by choosing a suitable axis to which the tail moments may be referred.

Assume the fuselage to be mounted as a cantilever from the wing. (Fig. 2) It is, in general, possible to establish a point "O" where it may be imagined that all fuselage flexibility is concentrated in the form of a spring-loaded hinge. The distance ($\bar{x}$) of this point from the tail reference axis is given by:

$$\bar{x} = - \frac{\gamma_L}{\gamma_M}$$

where γ_L, γ_M, and γ_N represent the changes in tail incidence contributed by fuselage flexibility when unit tail force, couple, or load factor is applied.

If aerodynamic tail moments are transferred to axis "O", it may be shown that:

$$C_{\bar{M}T} = \frac{\left[C_{\bar{M}Tr} + C_{\bar{M}\alpha T}\,\gamma_N\, N\right]}{\left[1 - C_{\bar{M}\alpha T}\,\gamma_M c_T q S_T\right]} \tag{17}$$

$$\alpha_T = \alpha_{Tr} \pm \left[\gamma_M\, C_{\bar{M}T}\quad c_T q S_T + \gamma_N\quad N\right] \tag{18}$$

where $C_{\bar{M}Tr}$ and $C_{\bar{M}T}$ represent the tail pitching moment coefficients about axis "O" with a rigid and flexible fuselage respectively while $C_{\bar{M}\alpha T}$ is slope of the tail pitching moment coefficient curve with respect to tail angle of attack and is, therefore, independent of fuselage flexibility. Thus, an analysis of the pitching moment characteristics of the airplane vs. angle of attack may first be made assuming the fuselage to be rigid. Eqs. (17) and (18) may then be used to add the effect of fuselage flexibility.

C - Trim, Stability, and Control

Eqs. (14), (15), (17), and (18) give all information necessary to compute trim conditions in the conventional manner.

Eqs. (15) and (17) may be used to transfer tail moments to airplane c.g;

$$\left(\frac{dC_{MTg}}{d\delta}\right)_{\text{rigid fuselage}} = \left[C_{M\delta T} + \frac{xg}{C_T}\, C_{L\delta T}\right] \tag{19}$$

$$\left(\frac{dC_{MTg}}{d\delta}\right)_{\substack{\text{flexible}\\ \text{fuselage}}} = \frac{\left[C_{M\delta T} + \left(\frac{\gamma_L}{\bar{c}_T \gamma_M}\right) C_{L\delta T}\right]}{\left[1 - \gamma_M \bar{c}_T q S_T \left\{ C_{M\alpha T} + \left(\frac{\gamma_L}{\bar{c}_T \gamma_M}\right) C_{L\alpha T}\right.\right]} - \left[\frac{xg}{\bar{c}_T} + \frac{\gamma_L}{c_T \gamma_M}\right] \tag{20}$$

where xg = distance from tail reference axis to airplane c.g.

The quantities $C_{M\alpha T}$, $C_{L\alpha T}$, $C_{M\delta T}$, and $C_{L\delta T}$ (where T indicates tail as opposed to wing) may be evaluated from expressions like those following Eq. (14). Elevator reversal may be obtained by plotting $(dC_{MTg}/d\delta)$ vs q.

The rolling moment due to aileron is given by Eq. (14) when the airplane centerline is used as the reference axis. Reversal speed may be obtained by plotting $(dCm/d\delta)$ vs q.

The "Rate of Roll Coefficient," pb/2V is equal to the quantity D appearing in Eq. (14) for a symmetrical flight. Since B, α, and N have no significance in this case, steady roll will occur when $C_M = 0$ and

$$\frac{pb}{2V} = D = -\frac{\delta C_{M\delta}}{C_{MD}}$$

The divergence speed may be obtained from Eq. (14) by setting equal to zero the denominator of the last term in the expression for any of the coefficients C_{MB}, $C_{M\alpha}$, etc. It is theoretically possible, but highly improbable, to obtain several different divergence speeds corresponding to the different "D" quantities. In any practical case there will be one for symmetrical and one other for unsymmetrical flight, and it may be presumed that $D_s = D_B = D_\alpha$, etc., corresponding to a single symmetrical "Characteristic Shape" and $D_u = D_\delta = D_D$, etc., for a single unsymmetrical "Characteristic Shape."

COMPARISON OF THEORY AND TEST

Comprehensive aeroelastic analyses and wind tunnel tests were conducted on a model of a typical high performance airplane with swept surfaces. The model is shown in Figure 3, and its principle specifications were as follows: span - 77.5", wing area - 750 in.^{2}, aspect ratio - 8, sweepback - 25%, chord line - 40°, airfoil perpendicular to 25% chord line - 65A-110.

The wing and horizontal tail were designed so as to be structurally similar to an actual full scale design. Thus, the model stiffness and weight were so distributed that nondimensional deflections under aerodynamic and gravity loads were the same as full scale under corresponding conditions. Fuselage flexibility was provided by incorporating a spring-loaded hinge at the reference axis as defined in Figure 2. Provision was made for clamping the fuselage. A dynamic pressure of $q = 61.5$ lbs/ft^2 on the model corresponded to full scale maximum design speed at sea level. The model was static tested carefully to obtain actual structural data, and these were used in the aeroelastic analysis.

Wind tunnel tests were carried out at the MIT Wright Brothers Wind Tunnel to determine longitudinal static stability, trim, and control characteristics. The test procedures were relatively conventional except that all runs were repeated at different values of q, and test data have been presented as functions of q. It was necessary to restrict the total lift force on the model at all times to a value corresponding to full scale design load factor since the flexibility and weight requirements automatically provided approximately the same relative strength as full scale. Thus, the maximum allowable lift force was 75 pounds, and this made it necessary for the balance to operate near its limit of sensitivity and has accounted for considerable scatter in the test points. This is an inherent difficulty of aeroelastic testing, and a specially designed low capacity sensitive balance would be desirable.

Rate of roll tests were conducted at the Cal Tech 10-foot tunnel by removing the tail and mounting the wing and fuselage from a sting.

The aeroelastic analysis was based upon the Weissinger (Reference 11) method of computing spanwise load distribution on a rigid wing and the Multhopp (Reference 12) method of treating control surface discontinuities. These procedures have been summarized and combined with uniform nomenclature and numerical tables of constant coefficients in Reference (9), which provides a convenient coordinated reference handbook for this work. In all cases seven control points were used over the entire span. Structural influence coefficients were used based upon structural design data in Reference (10).

Test data have been presented in Figures 4 through 7, which are self-explanatory. For longitudinal control and stability characteristics, the agreement between test and theory is substantially within the scatter of the experimental points. Appreciable discrepencies are present in the lateral case however. It is believed that this is largely due to the use of only 7 control points on the wing which provided only 2 points on each aileron. It is recommended that 15 or even 31 points be used for lateral control aeroelastic analysis.

CONCLUSIONS

The method presented herewith may be used to analyze any flexible structure, the loads on which depend linearly upon its deflection or position, provided that the structure could have been analyzed if deflections had been neglected. Thus, it is applicable to any steady state aeroelastic problem that does not involve serious nonlinearities such as may be encountered at the stall or at transonic speeds.

The method is systematic and well adapted to machine methods of computation. It does not require the assumption of arbitrary deflection shapes, and computational difficulties do not arise when conditions of instability or divergence are approached. As such, it may be used economically for a thorough and complete analysis of very flexible surfaces where aeroelastic effects are serious. More elementary approximate methods are preferable for rough estimates for the deflection on relatively rigid surfaces where aeroelastic effects may be assumed small.

The method permits complete evaluation of all tedious structural and aerodynamic operations before flight conditions such as dynamic pressure, load factor, angle of attack, etc., must be specified. Thus, many different flight conditions may be solved with relatively little effort once the basic parameters have been evaluated.

The concept of "Characteristic Shapes" greatly extends and simplifies the application of the method. This is particularly true since a specific "Characteristic Shape" will be associated with symmetrical flight conditions regardless of how the initial loading may have been applied, i.e., by change in angle of attack or by drooping of the flaps or due to deflections caused by load factor. The same applies for unsymmetrical flight.

Analytical results agree very well with test data for longitudinal stability and control; appreciable quantitative discrepencies occur for lateral conditions, however, when only 7 points are used on the span in setting up the air forces.

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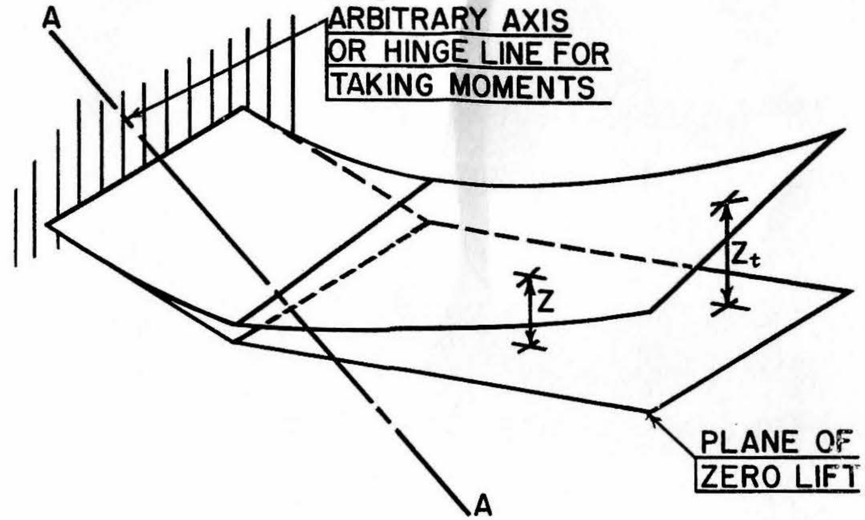

FIG.- 1
DEFLECTION COORDINATES

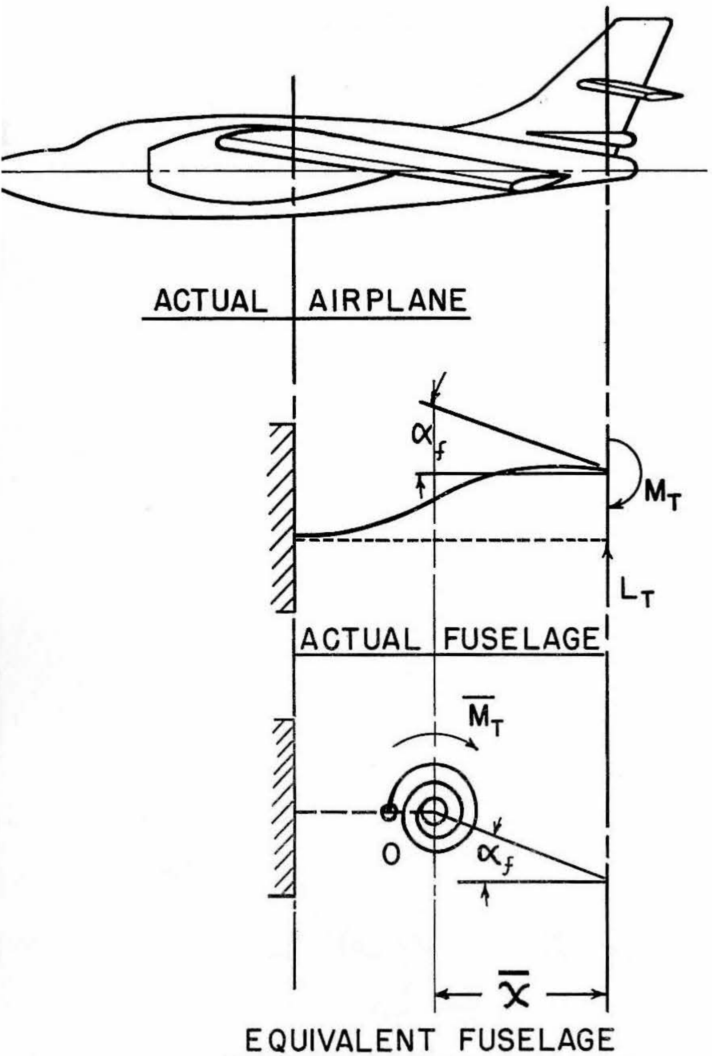

FIG.- 2
FUSELAGE REFERENCE AXIS

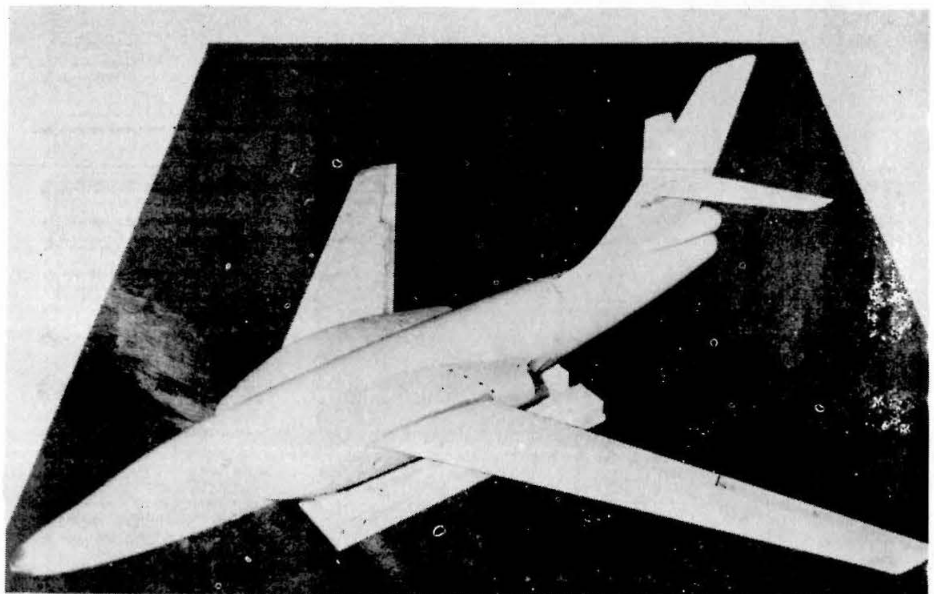

FIG.- 3

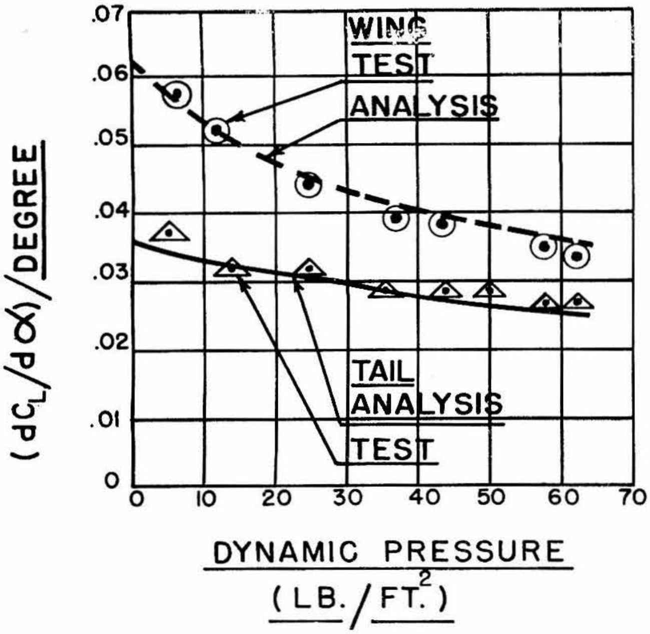

FIG.-4

SLOPE OF LIFT CURVE

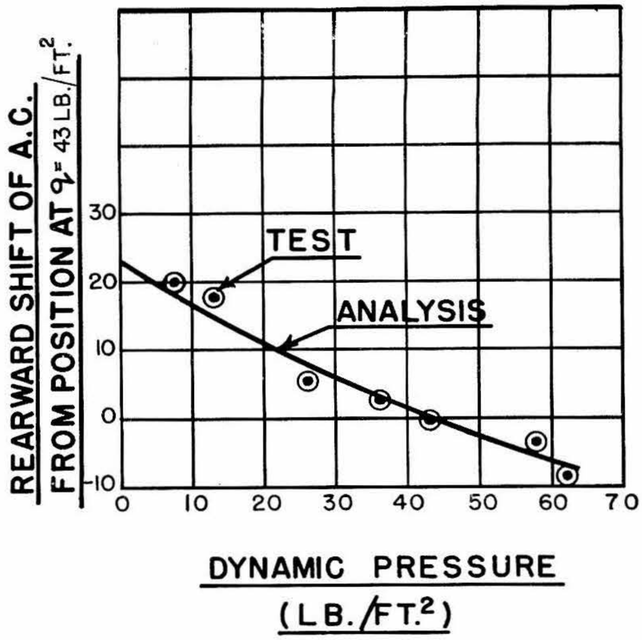

FIG.-5

SHIFT OF AERODYNAMIC CENTER

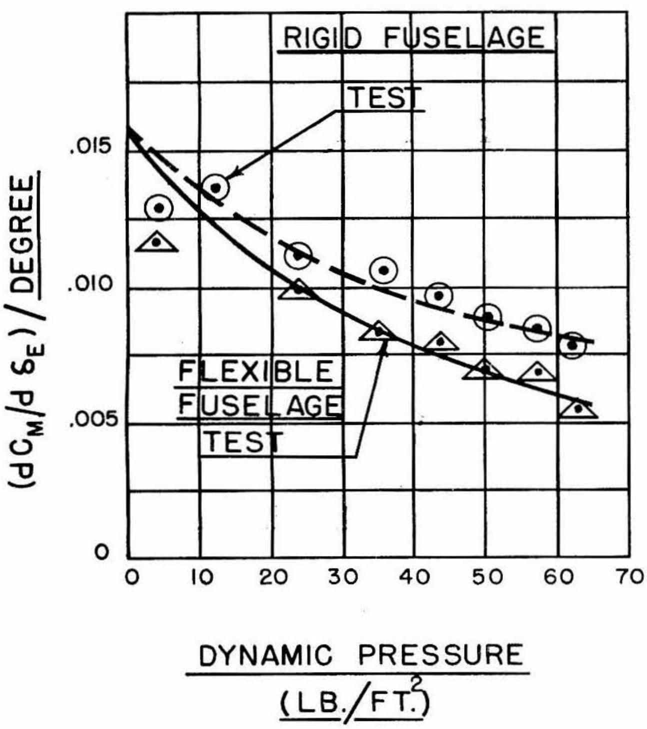

FIG.-6

ELEVATOR EFFECTIVENESS

(PITCH.MOM.COEFF./ELEV.ANGLE)

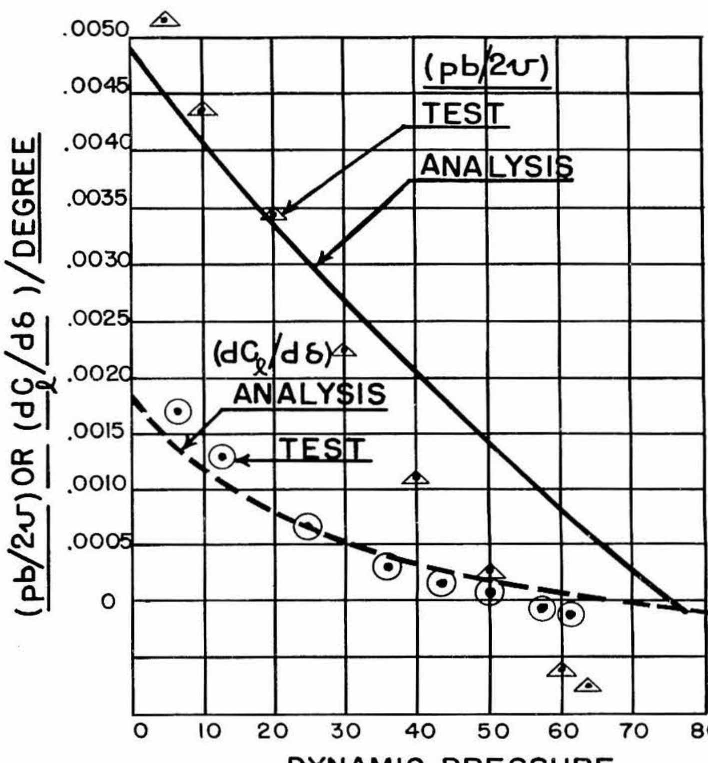

FIG.-7

AILERON STATIC EFFECTIVENESS AND RATE OF ROLL